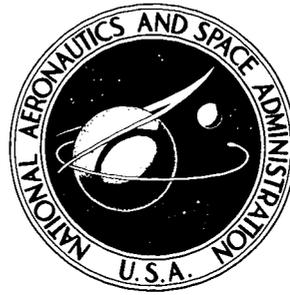


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# APOLLO EXPERIENCE REPORT - THERMAL PROTECTION SUBSYSTEM

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16. Abstract  <b>The Apollo command module was the first manned spacecraft to be designed to enter the atmosphere of the earth at lunar-return velocity, and the design of the thermal protection subsystem for the resulting entry environment presented a major technological challenge. Brief descriptions of the Apollo command module thermal design requirements and thermal protection configuration, and some highlights of the ground and flight testing used for design verification of the system are presented. Some of the significant events that occurred and decisions that were made during the program concerning the thermal protection subsystem are discussed.</b>			
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# APOLLO EXPERIENCE REPORT

## THERMAL PROTECTION SUBSYSTEM

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### SUMMARY

The Apollo thermal protection subsystem was designed to protect the command module during entry at lunar-return velocities. Discussed in this report are the major activities associated with the design, development, and flight testing of the subsystem and the significant technical and management decisions that evolved during the program. An extensive ground-based test development program in plasma arc heated facilities was conducted to characterize the ablator thermal performance, followed by full-scale thermal-vacuum tests at the NASA Lyndon B. Johnson Space Center and a comprehensive unmanned flight test program. Continuous efforts were made during the development of the Apollo thermal protection subsystem to reduce the weight of the subsystem; these efforts led to the reduction in ablator material density from 66 to 31 lb/ft<sup>3</sup>, but the total subsystem weight always trended upward. The thermal protection subsystem performed well on all operational lunar missions, and no anomalies requiring post-flight investigation were recorded. It is concluded that an adequate technology now exists to permit the efficient design of ablative heat shields for entry at lunar-return velocities.

### INTRODUCTION

The Apollo Program was the third in a series of manned space programs undertaken by the United States, and the Apollo command module (CM) was the first manned spacecraft designed to return to the earth from the moon. Project Mercury and the Gemini Program, which preceded the Apollo Program, were earth-orbital missions that resulted in atmospheric entry velocities of 26 000 ft/sec (inertial). The lunar-return trajectory of the Apollo spacecraft resulted in an atmospheric entry velocity of 36 333 ft/sec (inertial), which created an aerodynamic heating environment approximately four times as severe as that experienced by the Mercury and Gemini spacecraft. In addition, the deep-space and lunar environments imposed stringent thermal-control requirements on the Apollo spacecraft.

The precise definition of the entry thermal environment and the design of a heat shield to protect the entry module from the greatly increased heating environment were areas of primary technical concern in the design of the Apollo CM. A brief summary

of the design, development, and testing of the Apollo thermal protection subsystem (TPS) and a discussion of the significant technical and management decisions that evolved during the program are given in this report. The Apollo spacecraft flew on numerous earth-orbital and lunar-landing missions, and there were no problems or anomalies associated with the TPS. The success of the system can be attributed to the somewhat conservative design philosophy that was adopted and to the rigorous analytical and test certification requirements that were imposed. The report is concluded with some observations on design requirements and ablator performance.

## THERMAL PROTECTION DESIGN REQUIREMENTS

The Apollo CM TPS was designed to protect the CM during entry into the earth atmosphere at lunar-return velocities. The induced thermal environment resulting from such an entry necessitates the installation of a heat-shield material on the CM. Such material must be capable of sustaining (without excessive erosion) the temperatures caused by the high heating rates on the blunt face of the entry vehicle and must also provide insulation to minimize excessive substructure temperatures.

Early in 1961, a set of entry trajectories was developed by North American Rockwell (NR) that was based on a lift-to-drag ratio (L/D) of 0.5 at a trim angle of attack of  $33^\circ$ , a lunar-return speed of 36 333 ft/sec, and an entry along a flightpath inclined  $40^\circ$  to the equator. From this set of trajectories, two trajectories that formed a prescribed entry corridor were selected to size the heat shield. The two trajectories (fig. 1) initially selected for TPS design were called Block I. The overshoot boundary, designated HSE-3A, was limited to a 5000-nautical-mile range and used aerodynamic lift for a "skip-out" flightpath that maximized the heat load. The undershoot boundary, designated HSE-6, was of short duration to maximize the heating rate and was predicated on a 20g deceleration limit based on a biomedical constraint. In addition, the Block I design required that the ablator sizing include the effects of the heating environment during ascent flight (with no specific temperature limit imposed) and the thermal environment in space. In space, the temperature requirements for the ablator were limited to a cold temperature of  $-260^\circ\text{F}$  and a hot temperature of  $250^\circ\text{F}$ . The maximum initial bondline temperature at the start of entry was specified as  $250^\circ\text{F}$ , and the limiting temperatures at splashdown were  $600^\circ\text{F}$  at the interface of the ablator and the stainless-steel honeycomb structure and  $200^\circ\text{F}$  for the aluminum-honeycomb pressure-vessel structure.

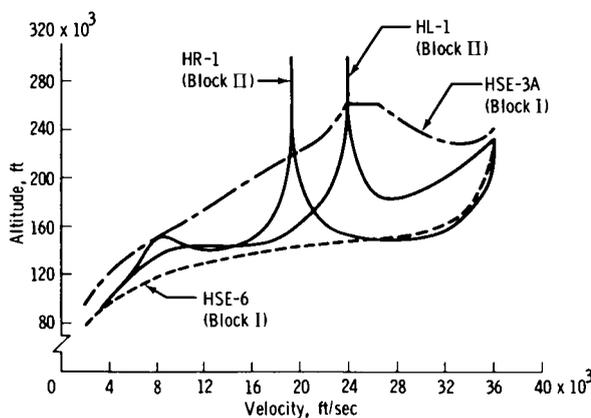


Figure 1. - Block I and II design entry trajectories.

In the fall of 1963, the development of the CM was such that numerous weight-saving refinements in the design and new operational logic in the program dictated the need for a design change. This change, called the Block II design, provided an opportunity to resize the TPS. A guidance

and control system had been established that could control the entry of the CM by lift modulation (in conjunction with a skip-out) to achieve the desired range. Packaging of the internal components in the CM to obtain a center-of-gravity offset sufficient for a 33° angle of attack could not be maintained, and the angle of attack was reduced to 28°. The requirement for ranging was reduced from a maximum of 5000 nautical miles to a maximum of 3500 nautical miles, which included 1000 nautical miles of maneuvering capability for weather avoidance. Because of the improved guidance system and reduced ranging requirements, the design entry boundaries were altered and a new set of design trajectories was generated. The new trajectories were based on a maximum L/D of 0.4, an initial entry angle of -5.20° to -9.45° to the horizon at 400 000 feet, and an increase in weight of the CM from 9500 to 11 500 pounds.

The Block II design trajectories are also shown in figure 1. The HL-1 overshoot trajectory skips out to a higher altitude than the corresponding Block I trajectory, but the entry time and the integrated heat load were reduced considerably. The HR-1 undershoot trajectory significantly increased the heating rate over that for Block I, but the corresponding heat-load change was insignificant. A summary comparison of the Block I and Block II entry design parameters is shown in table I. Additional Block II design requirements included a reduction in the ablator temperature extremes in space to ± 150° F (with a limit of 250° F for 30 minutes during lunar orbit), and initial bondline temperatures before entry of 150° F for the conical heat shield and 130° F for the aft heat shield. During entry, the bondline temperatures on the aft heat shield and crew compartment heat shield were limited to 600° F at any time before main parachute deployment, and the bondline design temperature of the forward compartment heat shield was limited to 600° F any time before jettisoning. The boost heating requirements for the heat shield were eliminated by the installation of a boost protective cover over the CM. The cover is jettisoned along with the launch escape tower 200 seconds after lift-off.

TABLE I. - APOLLO ENTRY DESIGN PARAMETERS

Parameter	Block I design		Block II design	
	Undershoot HSE-6	Overshoot HSE-3A	Undershoot HR-1	Overshoot HL-1
L/D	0.5	0.5	0.4	0.4
Maximum deceleration, g	20	NA <sup>a</sup>	20	NA
Entry velocity, inertial, ft/sec	36 333	36 333	36 333	36 333
Orbit inclination to equator, deg	40	40	40	40
Atmosphere properties	1959 ARDC <sup>b</sup>	1959 ARDC <sup>b</sup>	1962 standard <sup>c</sup>	1962 standard <sup>c</sup>
Maximum down-range distance, n. mi.	NA	5000	NA	3500
CM weight, lb	9500	9500	11 500	11 500

<sup>a</sup>Not applicable.

<sup>b</sup>Air Research and Development Corporation.

<sup>c</sup>U. S. standard atmosphere.

## THERMAL PROTECTION SUBSYSTEM DESCRIPTION

The external shape of the Apollo CM (fig. 2), like the Mercury (ref. 1) and Gemini spacecraft (ref. 2), consists of a blunt entry face with a conical afterbody that was designed to minimize convective heating during atmospheric entry. The center of gravity of the CM is offset from the axis of symmetry to generate the necessary lift to satisfy entry corridor and range requirements.

The TPS comprises the entire outer shell of the CM and consists of an ablator bonded to a stainless-steel structure that is fabricated in three subassemblies (fig. 3). In addition to protecting the CM from the thermal environment, the outer shell transmits the aerodynamic loads to the primary structure during boost and entry and transmits the hydrodynamic pressures to the primary structure during a water landing. Because of the uncertainties concerning the flow field during CM entry into the atmosphere of the earth, the magnitude of radiation heating, and the analysis of ablator-to-metal junctions, the decision was made to fabricate the entire CM TPS from ablative material. This decision was made, even though temperature predictions indicated that reradiative metal shingles would provide sufficient thermal protection for much of the CM conical afterbody.

The ablative material selected for the TPS is designated Avco 5026-39G and consists of an epoxy-novalac resin reinforced with quartz fibers and phenolic microballoons. The density of this material is  $31 \text{ lb/ft}^3$ . The ablator is applied in a honeycomb matrix that is bonded to a stainless-steel substructure. The phenolic honeycomb is first bonded to the stainless-steel shell with HT-424 adhesive, and then the ablator is inserted into the individual honeycomb cells with a hypodermic device that is similar to a caulking gun.

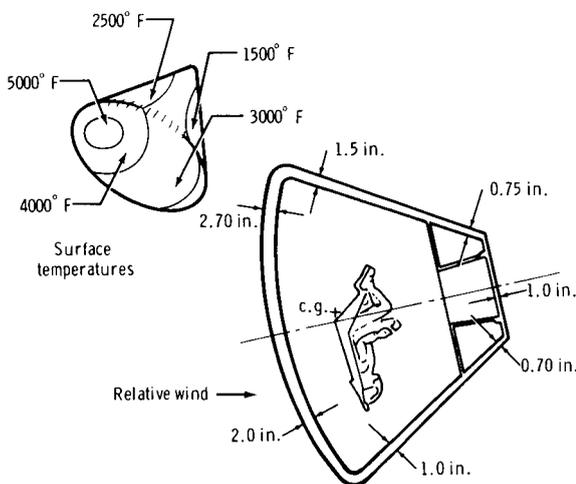


Figure 2. - Command module ablator thickness.

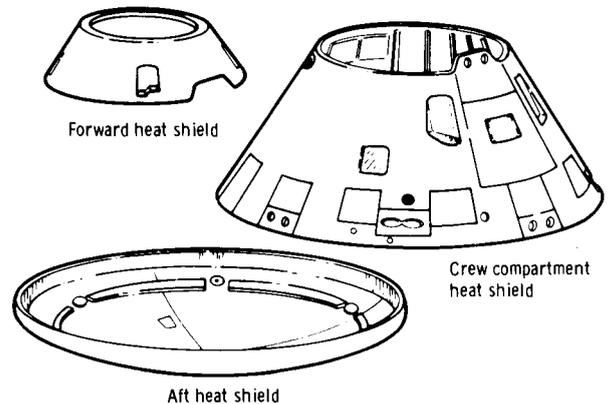


Figure 3. - Heat-shield substructure assemblies.

The thickness of the ablator varies with the local thermal environment (ref. 3) and corresponding temperature profile, as shown in figure 2. Two typical cross sections of the TPS and primary structure are shown in figure 4. Section A-A represents the stagnation heating area where the total heat load is a maximum and requires an ablator thickness of 2.7 inches. Section B-B cuts through the leeward side where the heating rates are lowest and the ablator thickness is 0.7 inch. The space between the outer TPS shell and the cabin structure is filled with a low-density ( $3.5 \text{ lb/ft}^3$ ) fibrous insulation, TG15000. This insulation is used to reduce the heat transfer between the outer shell and the cabin wall during space flight and, in particular, during entry into the earth atmosphere.

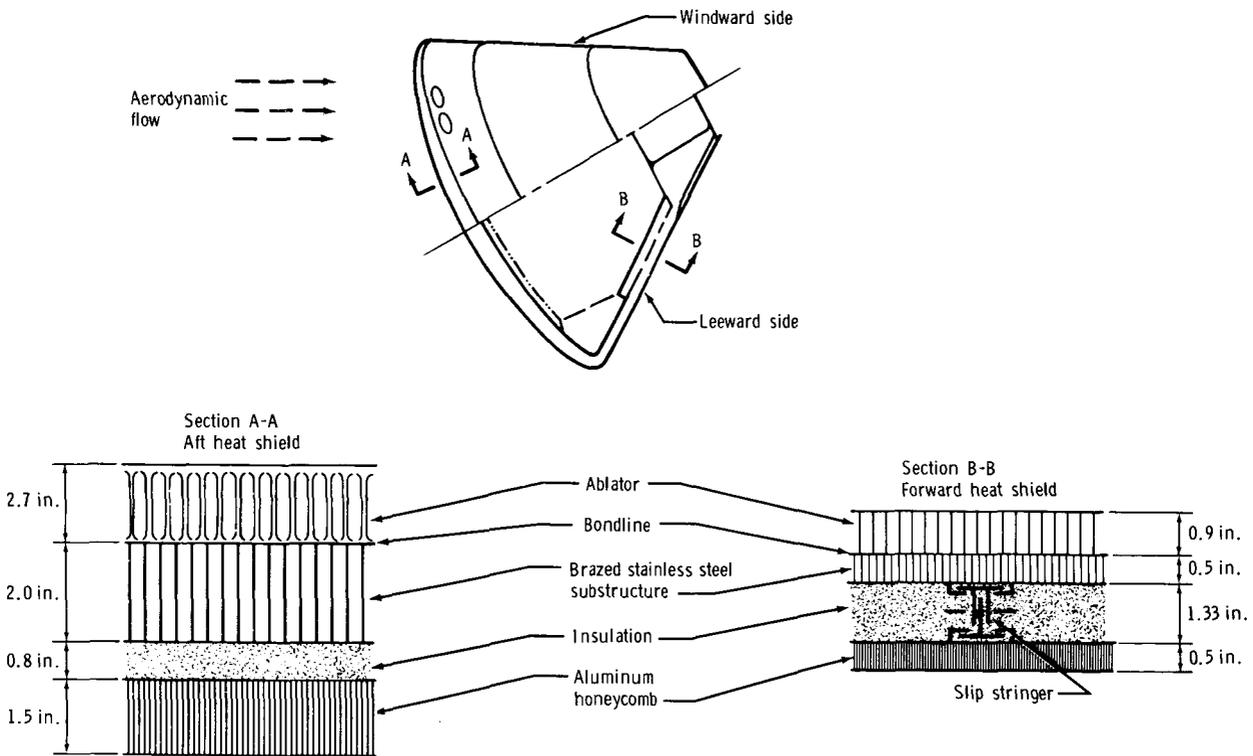


Figure 4. - Structural arrangement of Apollo TPS.

To accommodate the heat-shield deformations that occur because of the thermal extremes in space and entry heating, the conical section of the heat shield is attached to the aluminum cabin structure by means of a system of fiber-glass slip stringers. This attachment system (fig. 4) provides strain isolation between the inner and outer structures and reduces heat conduction from the heat shield to the cabin. The thermal control requirements for the spacecraft in outer space necessitates a relatively low thermal absorptance-to-emittance ratio of 0.4 for the surface of the CM. This low ratio is achieved with a pressure-sensitive Kapton polyimide tape that is coated with aluminum and oxidized silicon monoxide and that is applied over the entire external

surface of the ablator. The installation of a boost protective cover over the conical portion of the CM prevents contamination of the thermal-control coating and the CM windows by aerodynamic heating during boost and by the tower jettison engine plume. The boost protective cover, which is attached to the launch escape tower and is jettisoned with the tower before orbital insertion, consists of a layer of cork bonded to a fiber-glass cloth backing. Details of the boost protective cover are shown in figure 5 and are discussed in reference 4.

In addition to the basic thermal environment design considerations, the Apollo heat shield also has numerous penetrations and protuberances for the installation of components such as windows, reaction control engines, antennas, and vents, as shown in figure 6. Each of these discontinuities in the TPS required special design considerations such as the recessing of the components and the use of densified ablators in local adjacent areas.

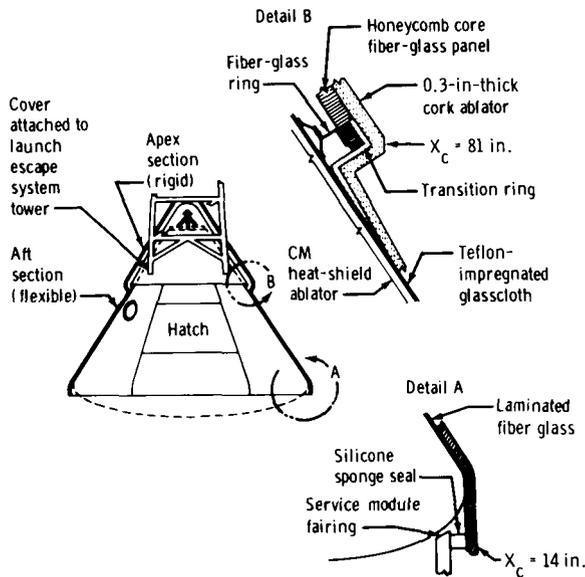


Figure 5. - Boost protective cover for Apollo command module.

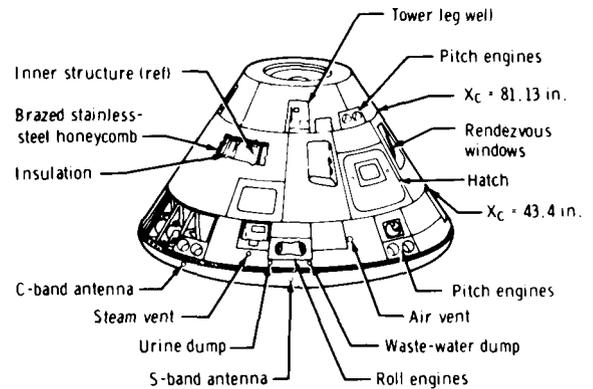


Figure 6. - Penetrations in command module heat shield (Block II).

## FABRICATION

The Apollo TPS consists of an ablator in a honeycomb matrix bonded to a stainless-steel substructure. The substructure is made up of three subassemblies (fig. 3), which are referred to, respectively, as the aft heat shield, the crew compartment heat shield, and the forward heat shield. Each subassembly contains several brazed sandwich panels that are welded together by NR using a tungsten inert gas process. A typical weld-assembly sequence for the forward compartment heat shield is shown in figure 7. This subassembly consists of four large brazed panels and four tower-well fittings, which are first welded together and to which the forward ring and panel and the aft ring are added to complete the assembly.

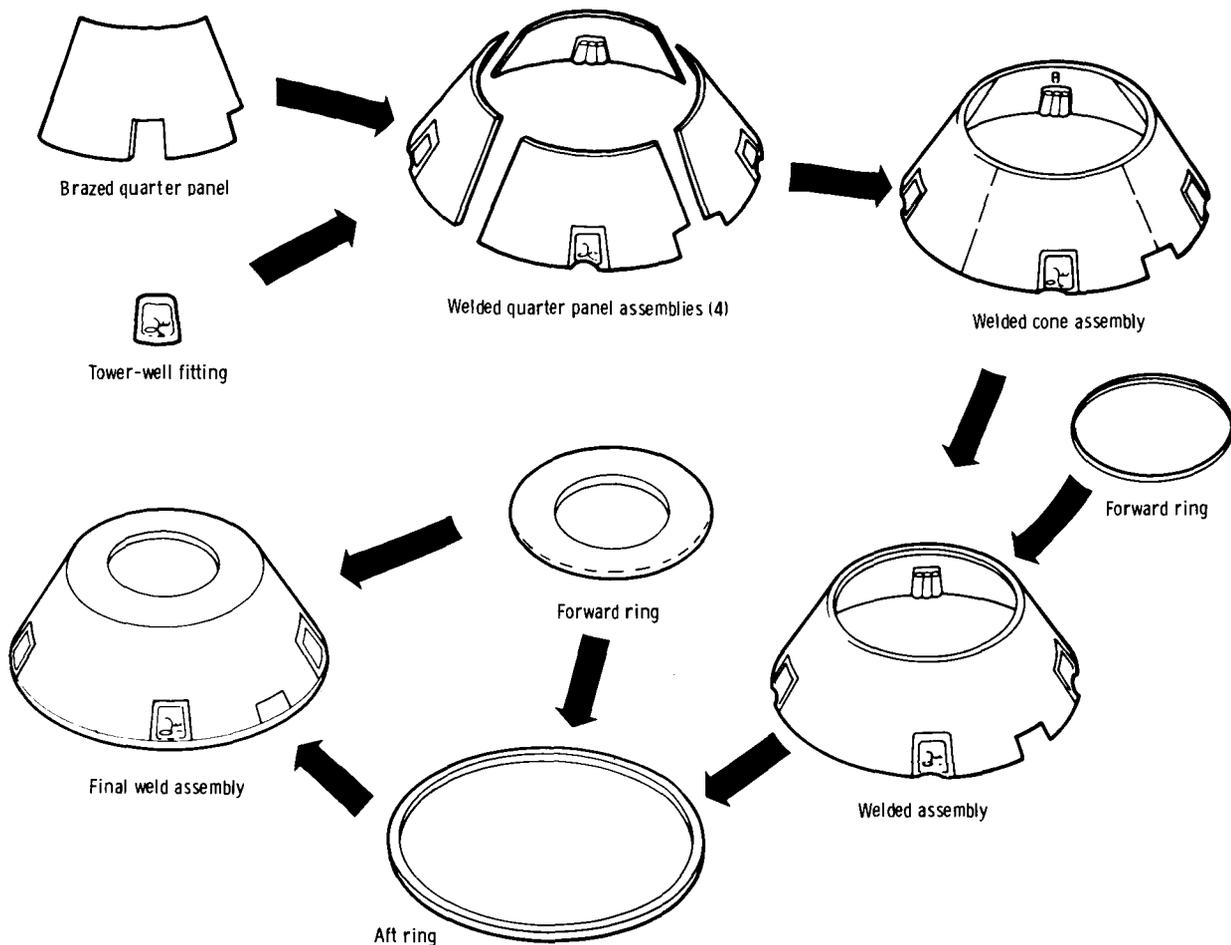


Figure 7. - Fabrication of forward heat shield substructure.

A total of 41 brazed sandwich panels constitutes a shipset for each CM. These panels were manufactured by the Aeronca Corporation under subcontract to the NR Corporation. The first several shipsets were made of PH15-7MO stainless steel; however, the later shipsets were constructed of PH14-8MO stainless steel because of the better cryogenic toughness of the 14-8 material.

After the panels are welded by NR, the three subassemblies are sent to the Avco Corporation for application of the ablator. The structure is first cleaned by scrubbing with an abrasive detergent slurry, and a primer coating is applied before the bonding of the fiber-glass honeycomb with HT-424 tape adhesive. The fiber-glass honeycomb core sections are then fitted in place over the tape, and the edge members are positioned at the same time. The assembly is vacuum bagged, and the adhesive is oven cured at 325° F for 1 hour. Inspection of the bonding of the honeycomb to the structure is made by a nondestructive ultrasonic transmission evaluation. Any unbonded areas are repaired, and then the assembly is ready for application of the ablator into the honeycomb. This operation, termed "gunning," is the injection of the Avcoat 5026-39G into each cell of the honeycomb by means of a special gun developed for that purpose. The cylindrical cartridges containing the ablator are dielectrically heated to 160° F and are inserted in the gun. When the nozzle is positioned over the honeycomb cell, a

solenoid-controlled air valve injects a blast of air into the cartridge and this entrains the ablator and carries it into the cell, filling it from the bottom to the top. There are approximately 370 000 cells in the honeycomb. A photograph of the gunning of the honeycomb cells on the forward heat shield is shown in figure 8. When all the cells are filled, the assembly is vacuum bagged and the ablator is oven cured for 16 hours at 200° F; then, it is postcured for an additional 16 hours at 250° F. Then, the entire surface is machined on a numerically controlled turret lathe to the design-thickness requirements. The thickness of the ablator is measured by an eddy-current technique at preselected points, during machining as a process control, and after machining as a final acceptance measurement. The machined TPS is radiographed to detect any defects in the ablator, and repairs are made if necessary. Then silicone rubber gaskets are inserted in all door openings, and various details (such as bolt plugs, molded ablator parts for the abort-tower wells, and fiber-glass shear and compression pads) are bonded in place. After completion of these operations, the main ablator is checked for moisture content. A layer of thin, epoxy-based pore sealer and a moisture-protective plastic coating then are applied to the surface to ensure sealing of the porous ablator.

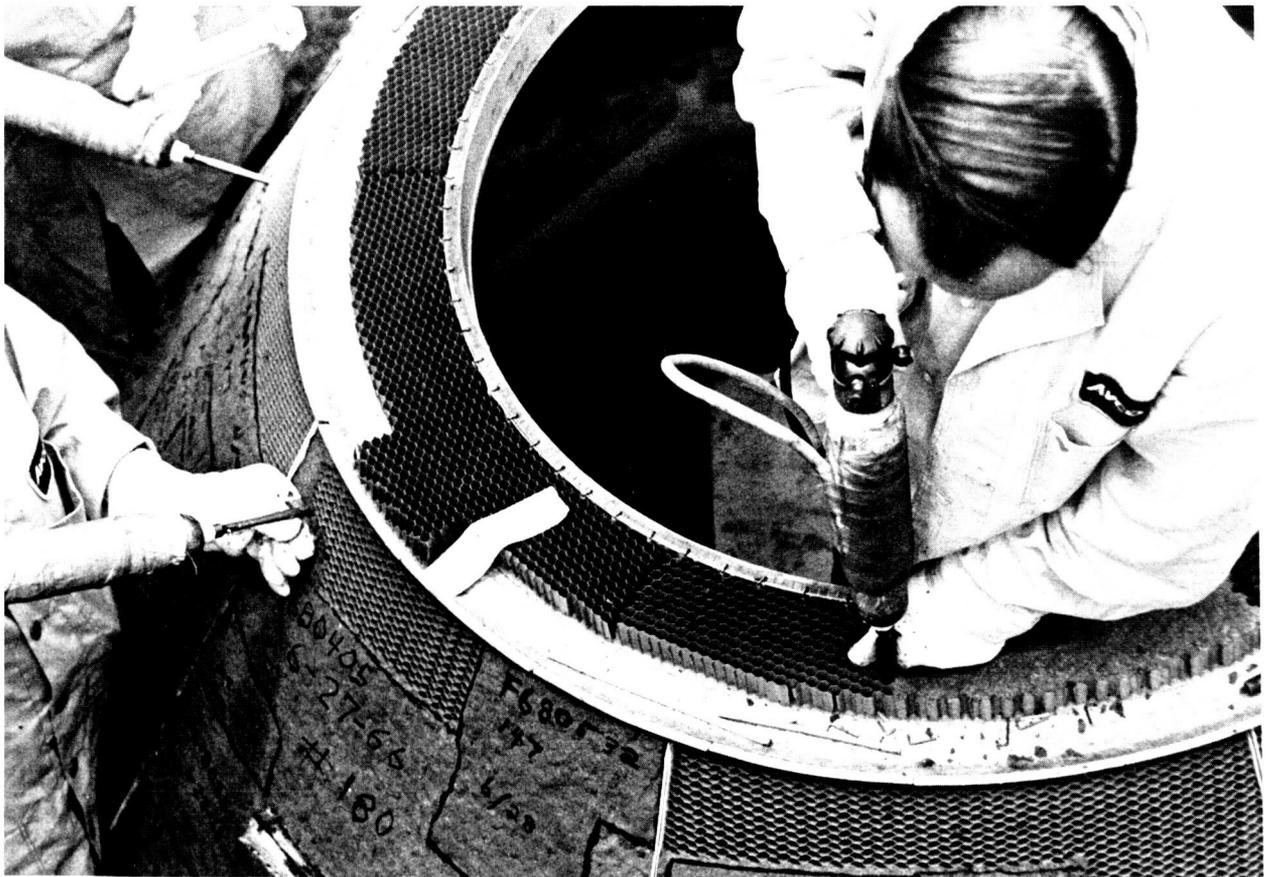


Figure 8. - Injection of ablator into honeycomb cells.

After this operation, the final weight and center-of-gravity measurements are made, and the heat-shield subassemblies are returned to NR for installation on the spacecraft. Before the CM is shipped from the prime contractor site to the NASA John F. Kennedy Space Center (KSC), the plastic coating is stripped off and the thermal-control coating with an adhesive backing is attached to the CM.

## GROUND AND FLIGHT VERIFICATION TESTING

In conjunction with detailed thermal and structural analyses, the Apollo spacecraft heat shield was certified for manned operations by means of an extensive ground and unmanned flight test program. Three full-scale command modules (CM 004, CM 008, and 2TV-1) were used for the ground test program. The CM 004 vehicle (without the ablator, but complete in all other structural and thermal respects) was subjected to a radiant-lamp heating test that duplicated, in real time, the predicted bondline temperatures of the heat shield for the Block I overshoot design entry trajectory. The objectives of this test (February 1966) were to evaluate the thermal-structural behavior of the TPS substructure (including the windows and hatches and the strain-isolation attachment system between the heat-shield substructure and the cabin structure) and to verify the thermal capability of the TF15000 insulation to limit the temperature of the aluminum wall of the cabin to a maximum of 200° F.

Thermal-vacuum testing of spacecraft 008 and 2TV-1 was conducted in the 65-foot-diameter thermal-vacuum chamber at the NASA Lyndon B. Johnson Space Center (JSC). A photograph of spacecraft 2TV-1 inside this chamber, with one side of the spacecraft

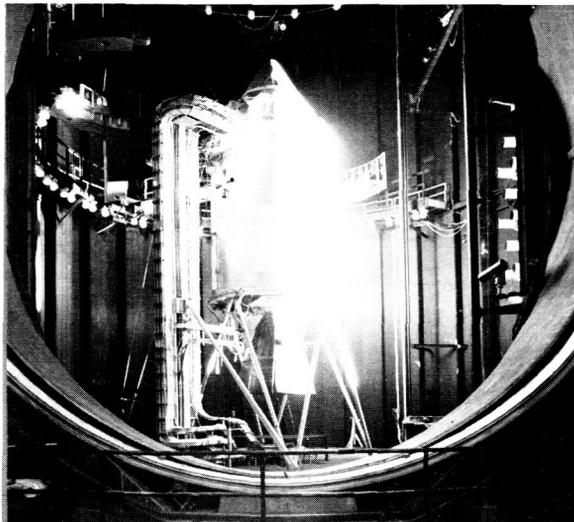


Figure 9. - Space-flight-mission simulation test (spacecraft 2TV-1) in thermal-vacuum chamber at JSC.

lighted by the simulated sun, is shown in figure 9. These tests subjected the spacecraft to the temperature extremes and the vacuum conditions expected in space flight. The solar thermal energy was simulated with carbon-arc heaters (fig. 9), and liquid nitrogen in the chamber wall provided deep-space cold simulation. These tests provided data on the expansion and contraction of the gaps between the heat-shield compartments, the integrity of the ablator when cold soaked, and a quantitative evaluation of the distortion of the crew compartment heat shield.

Flight test verification of the Apollo TPS was conducted on four unmanned spacecraft. The test flight parameters are given in table II. The first two Apollo test flights demonstrated the performance of the TPS during entry into the earth atmosphere from earth orbit. The first of these flights, mission AS-201, used the Saturn IB launch vehicle with spacecraft 009. The unmanned spacecraft was launched from KSC (February 26, 1966) on a suborbital ballistic flight that gave an entry inertial velocity of 26 482 ft/sec at a 400 000-foot altitude.

TABLE II. - APOLLO TPS FLIGHT VERIFICATION

Variable	Mission <sup>a</sup>			
	AS-201	AS-202	AS-501	AS-502
General information				
Booster	Saturn IB	Saturn IB	Saturn V	Saturn V
Spacecraft	009	011	017	020
Date launched	Feb. 26, 1966	Aug. 25, 1966	Nov. 9, 1967	Apr. 4, 1968
Mission description	Supercircular entry with high heat rate	Supercircular entry with high heat rate	Supercircular entry at lunar-return velocity	Lower velocity because of launch-vehicle malfunction
Entry conditions				
Inertial velocity at 400 000 ft, ft/sec . . . . .	26 482	28 512	36 545	32 830
Relative velocity at 400 000 ft, ft/sec . . . . .	25 318	27 200	35 220	31 530
Inertial flightpath angle at 400 000 ft, deg . . . . .	-8.60	-3.53	-6.93	-5.85
Range flown, n. mi. . . . .	470	2295	1951	1935
Entry time, sec . . . . .	674	1234	1060	1140
Maximum heating rate, Btu/ft <sup>2</sup> -sec . . . . .	164	83	425	197
Total reference heating load, Btu/ft <sup>2</sup> . . . . .	6889	20 862	37 522	27 824

<sup>a</sup>All missions were unmanned.

The entry trajectory for this first test flight was chosen to provide the highest heating rates and, consequently, the highest ablator-surface temperatures and surface-recession rates that could be achieved with the Saturn IB booster. The TPS performed well during this mission and qualified the subsystem for high-heating-rate entries from earth orbit.

The second test flight was mission AS-202, which again used a Saturn IB launch vehicle. This unmanned spacecraft was launched from KSC on August 25, 1966, for a suborbital flight with an entry trajectory designed to give the maximum total heat load that could be achieved from earth orbit. This flight qualified the TPS for this type of atmospheric entry. In addition to qualifying the system for manned earth-orbital entries, the AS-201 and AS-202 missions provided data for correlation with the thermal analytical models used for thermal-performance predictions at lunar-return entry velocities.

The Apollo 4 mission (also known as mission AS-501) was the first flight test involving a Saturn V launch vehicle with a lunar module test article (LTA-10R) and a Block II configured CM (spacecraft 017). The unmanned spacecraft was launched from KSC on November 9, 1967, for a planned flight time of 8 hours 37 minutes. After two revolutions in earth orbit, the S-IVB stage was reignited for a simulated translunar

injection burn. Shortly after the spacecraft separated from the S-IVB stage, the service propulsion subsystem was ignited for a short-duration burn to propel the spacecraft to an apogee altitude of 9769 nautical miles. For approximately 4.5 hours during the coast phase, the spacecraft was oriented with the CM hatch window pointed directly toward the sun. This attitude cold soaked the thick ablator on the side opposite the hatch and achieved the maximum thermal gradient around the CM heat shield. After the cold soak, the service propulsion subsystem was reignited for a long-duration burn to accelerate the CM to entry conditions that represented the most severe combinations of heating rate and heat load for the two extreme operational conditions that could possibly be achieved from a lunar-return trajectory. The entry trajectory resulted in an inertial velocity at 400 000 feet of 36 545 ft/sec and a trim L/D of 0.365. The CM landed in the Pacific Ocean within 10 miles of the predicted landing point, 1951 nautical miles down range from the entry interface at 400 000 feet. A comparison of the entry heating conditions of this test flight (AS-501) with those expected during manned operational flights is shown in figure 10.

The actual mission flown was very close to that planned, and postflight inspection of the recovered Apollo 4 CM indicated that the Block II TPS survived the simulated lunar return entry environment satisfactorily. Sufficient flight data were obtained to permit a thorough evaluation of the thermal performance of the Block II TPS. Temperature data were within design limits for the flight. The response from the extensive instrumentation was good, except for the heating rate measurements on the aft heat shield. Although the aft ablative heat shield was heavily charred and temperature data indicated surface temperatures approaching 5000° F, the measured surface recession was less than had been expected (based on ground test data) over all points on the aft heat shield.

The Apollo 6 mission (also known as mission AS-502) was the second mission to use a Saturn V launch vehicle with a lunar module test article (LTA-2R) and a Block II configured CM (spacecraft 020). This mission provided one further test of the TPS at lunar-return velocity. The only difference between the Apollo 6 mission and the Apollo 4 mission in regard to thermal configuration was that CM 020 had the Block II thermal-control coating removed and it was to be completely cold soaked during the coast phase. The only CM change made for the Apollo 6 mission was that, for the first time, the unified crew hatch would be a part of a CM undergoing test flight.

The unmanned spacecraft was launched from KSC on April 4, 1968, for a planned flight time of 9 hours 57 minutes. Because of a malfunction of the S-IVB in earth orbit, the service propulsion subsystem had to be used to achieve the programmed apogee of 12 000 nautical miles. Because of the resulting low fuel availability, the second firing of the service propulsion subsystem was inhibited and the CM achieved an inertial velocity of only 32 830 ft/sec at the entry interface of 400 000 feet. The heating conditions

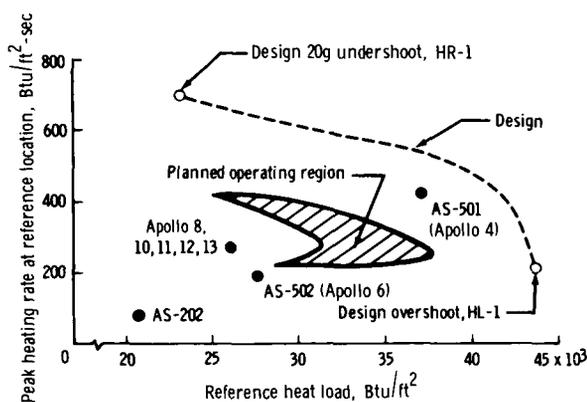


Figure 10. - Entry test conditions for flight test verification of Apollo CM heat shield.

achieved on this flight are shown in table II and figure 10. It can be seen that the referenced heating rate was only about half that experienced on the Apollo 4 flight, and the total referenced heat load was 10 000 Btu/ft<sup>2</sup> less.

The unmanned flights provided test verification of the Apollo TPS for both earth-orbital and lunar-return missions. The measured data obtained from these flights and from the first two manned flights (AS-205 and AS-503) were used to correlate the analytical models used for the required certification analysis.

A summary of the actual entry conditions for the manned lunar landing missions (Apollo 8 to 16 missions) is given in table III. As indicated in the summary, the maximum down-range entry distance was 1500 nautical miles compared with the established Block II design requirement of 3500 nautical miles. These results indicated a crew preference for a shorter down-range distance. The shorter down-range entry distance resulted in a maximum integrated heat load of 26 500 Btu/ft<sup>2</sup>, which is appreciably less than the design requirement of 44 500 Btu/ft<sup>2</sup>.

TABLE III. - SUMMARY OF ENTRY CONDITIONS FOR OPERATIONAL LUNAR MISSIONS

Mission/vehicle	Entry velocity, relative, ft/sec	Entry velocity, inertial, ft/sec	Entry angle, inertial, deg	L/D	Range, n. mi.	Entry time, sec	Reference $\dot{q}$ , Btu/ft <sup>2</sup> -sec (a)	Reference Q, Btu/ft <sup>2</sup> (b)
Apollo 8/103 AS-503	35 000	36 221	-6.48	0.300	1292	868	296	26 140
Apollo 10/106 AS-505	34 968	36 314	-6.542	.305	1295	871	296	25 728
Apollo 11/107 AS-506	35 024	36 194	-6.483	.300	1497	929	286	26 482
Apollo 12/108 AS-507	34 956	36 116	-6.50	.309	1250	815	285	26 224
Apollo 13/109 AS-508	34 884	36 211	-6.49	.2905	1250	835	271	25 710
Apollo 14/110 AS-509	34 996	36 170	-6.37	.280	1234	853	310	27 111
Apollo 15/111 AS-510	34 928	36 096	-6.51	.290	1184	778	289	25 881
Apollo 16/112 AS-511	35 502	36 090	-6.49	.286	1190	814	346	27 939

<sup>a</sup>Reference heating rate.

<sup>b</sup>Reference heat load.

## SIGNIFICANT EVENTS AND DECISIONS

### Thermal Protection Subsystem Fabrication Concepts - Tiles Compared With Monolithic Ablator

The TPS concept submitted initially by NR to design and manufacture the Apollo spacecraft consisted of ablative tiles made from phenolic-nylon material bonded to a

honeycomb-sandwich substructure made of aluminum. The substructure was to be built by the prime contractor, and the design, fabrication, and installation of the ablative tiles were to be accomplished by a subcontractor. In April 1962, a subcontractor was chosen to supply an ablative system consisting of molded tiles (typically 1-foot square) of Avcoat 5026-22 ablator bonded to a stainless-steel substructure. At approximately the same time (April 1962), recovered heat shields from Project Mercury were found to have experienced debonding of the tiled ablative center plug. This fact, together with the uncertainty regarding the thermodynamics at the joints between the tiles, led to a general lack of confidence by both NASA and NR in the tile method of ablator application. Consequently, NASA instructed NR to conduct an alternative fabrication study of the ablator installation method being demonstrated successfully at that time on the Gemini spacecraft. The Gemini heat shield consisted of a fiber-glass honeycomb core filled with an elastomeric ablator. As a result of this study, a lower viscosity Avco ablator (designated Avcoat 5026-39) was developed that could be applied in a monolithic fashion to a phenolic fiber-glass honeycomb having a cell size of 3/8 inch. The fiber-glass honeycomb was first bonded to the stainless-steel substructure with HT-424 adhesive, and the individual honeycomb cells were then filled with the ablator. Initially, the cells were filled with the ablator by tamping the dry ablator into the open cells, then curing the entire TPS installed on the vehicle. The tamping operation, however, caused considerable concern with respect to quality assurance and the possibility of damaging the substructure. Finally, the ablative material composition was modified so that it could be gunned in a mastic form (fig. 8) into the honeycomb cells. Although the monolithic ablator in a honeycomb matrix did provide a desirable fail-safe feature, it also resulted in longer manufacturing schedules and required additional inspection procedures.

## Material Selection for Heat-Shield Substructure

Stainless steel was chosen in preference to aluminum for the TPS substructure because of the fail-safe characteristics provided by a higher-melting-point alloy in the event of a localized loss of the ablator. The PH15-7MO alloy was the alloy originally proposed by NR because of its high tensile strength ( $F_{tu} \approx 200\,000$  psi at room temperature) and brazing compatibility. The initial heat shields were fabricated from this alloy. However, further investigations revealed that the material became brittle at low temperatures and the fracture toughness was unacceptable. The temperature criterion at this time for spacecraft during space flight was  $\pm 250^\circ$  F. Because of this fact, another material with better fracture toughness at  $-250^\circ$  F was sought and the alloy PH14-8MO (vacuum melted) was selected to replace PH15-7MO. The PH14-8MO exhibited outstanding fracture toughness throughout the temperature range of  $-250^\circ$  to  $600^\circ$  F. However, it was a relatively new alloy and an extensive development period was required to define the optimum welding and brazing process specifications.

## Thermal Protection Subsystem Weight History

The ablative material initially selected in April 1962 for the Apollo TPS was Avcoat 5026-22, which had a density of  $66 \text{ lb/ft}^3$ . The predicted TPS weight with this material was 1684 pounds. Shortly thereafter, improvements (which included the addition of microballoons) were made to the material so that, by the end of 1962, a

low-density version of the material (designated Avcoat 5026-39, with a density of 35 lb/ft<sup>3</sup>) was incorporated in the TPS. This represented a density reduction of 47 percent, but the corresponding reduction in predicted system weight was only 20 percent. The low rate of system weight reduction was caused by the inclusion of additional requirements (primarily the boost heating environment) which had been overlooked during the initial design phase. In the years that followed (after 1962), some further minor improvements were made in the ablative material, which culminated in a material density of 31 lb/ft<sup>3</sup>; but the predicted system weight for the TPS continued to have an upward trend (fig. 11). The upward weight trend and the causes (which have been true historically of all aircraft and spacecraft projects) can be attributed to the following factors.

1. The continually increasing number of protuberances on the outside moldline of the vehicle and a resulting increase in the local heating environment
2. The more refined analytical techniques that replaced earlier gross predictions
3. The addition of more rigorous thermal-control criteria as the spacecraft program progressed

Because of management concern about the increasing spacecraft weight, an attempt was made in 1964 to reduce the spacecraft weight. The Block II design, which resulted from these changes, showed a decrease in TPS weight of approximately 200 pounds (fig. 11). This was achieved by (1) the elimination of the effects of boost heating environment by the introduction of a boost protective cover that was jettisoned with the launch escape tower (fig. 5), (2) the reduction in the down-range requirement from 5000 to 3500 nautical miles (which provided a more realistic operational requirement), (3) a reduction in the maximum initial entry temperature from 250° to 150° F (by the use of an external thermal-control coating), and (4) the removal of some protuberances.

Two other weight-saving modifications to the TPS were incorporated in 1968 and were based on recommendations by NASA. The first modification was the removal of several layers of nylon from the soft insulation blanket installed between the TPS substructure and the aluminum cabin. The removal did not compromise the thermal insulation performance, was accomplished without causing any schedule delays, and resulted in a weight saving of 64 pounds. The second recommendation (a simple manufacturing change) eliminated the application of the protective enamel paint that acted as a moisture barrier. Because the thermal-control coating subsequently

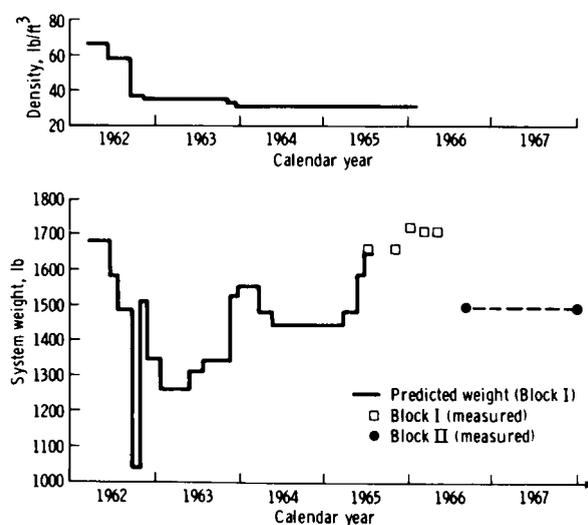


Figure 11. - History of ablator density and TPS weight.

applied to the ablator surface was also a good moisture sealer, an extra coating of thin epoxy-based pore sealer was applied instead of the enamel. This sealer, together with the thermal-control coating, was sufficient to keep the moisture content in the ablator below the specified 2 percent.

## Experimental Research Flights

The test verification plans for the Apollo TPS included full-scale flight testing, which began in 1966. To gain confidence in the thermal prediction and design methods being used, the NASA Langley Research Center (LRC) proposed two flight-research experiments. The first experiment was the Flight Investigation of Reentry Environment (FIRE) project, the objective of which was to measure quantitatively the convective and radiative heating environment on a subscale (2-foot maximum diameter) Apollo configuration at the correct lunar-return entry velocity. The flight was justified because of the many approximations inherent in the aerothermodynamic theories and to substantiate the magnitude of the thermal-radiation contribution to the total heating environment. The FIRE payload was launched from KSC on April 14, 1964, on a ballistic trajectory that resulted in an entry velocity (at a 400 000-foot altitude) of 37 900 ft/sec. Measurement sensors included beryllium heat-sink-type calorimeters for measuring the total (convective plus radiative) heat flux and radiometers for measuring the thermal radiation from the shock layer. The data from these sensors were relayed to earth by delayed telemetry after the radio-frequency blackout period. Although there were a few flight anomalies, with respect to perturbation in the body motion, the FIRE experiment was a success (ref. 5). A comparison of the measured heating data on the spherical forebody of the vehicle with several different theoretical predictions is shown in figure 12. These data provided confidence in the methods used for calculating the heating rates around the Apollo entry module.

The second subscale experimental flight in support of the Apollo TPS development was also proposed by LRC and had as its objective the verification of ablation performance at the correct enthalpy and heating rates (corresponding to the Apollo undershoot design (20g) trajectory). The justification for the experiment was that the high enthalpies associated with the Apollo entry trajectory could not be duplicated in ground test facilities. An added fifth stage for the Scout booster was being developed by LRC, and the five-stage Scout vehicle was proposed for the launching of the test spacecraft (known as the R-4). The entry nosecone used for the experiment was a spherical dish with a spherical radius of 17.4 inches and a diameter of 11.1 inches. The entry nosecone consisted of Avcoat 5026-39G ablator (1.25 inches thick) bonded to a stainless-steel structure, which simulated the Apollo aft heat shield (fig. 13). Instrumentation consisted of ablation sensors and thermocouples. The R-4 spacecraft was launched successfully by a five-stage Scout booster from Wallops Island, Va., on August 18, 1964. To match the Apollo heating rates at less than lunar-return entry velocity, the fifth stage of the Scout vehicle was ignited late in the entry phase into the earth atmosphere, so that high heating rates occurred at a lower altitude. At the lower altitude, the heating rates approximated those an Apollo vehicle would experience; however, the resulting free-stream dynamic pressures on the Scout vehicle were three times higher than the pressures an Apollo CM would undergo on entry into the earth atmosphere. The telemetered ablation-rate data from this flight indicated that the rates encountered were much higher than had been expected, particularly during the later stages of entry, and, in fact, resulted in complete erosion of the 1.25-inch-thick ablative material. There was

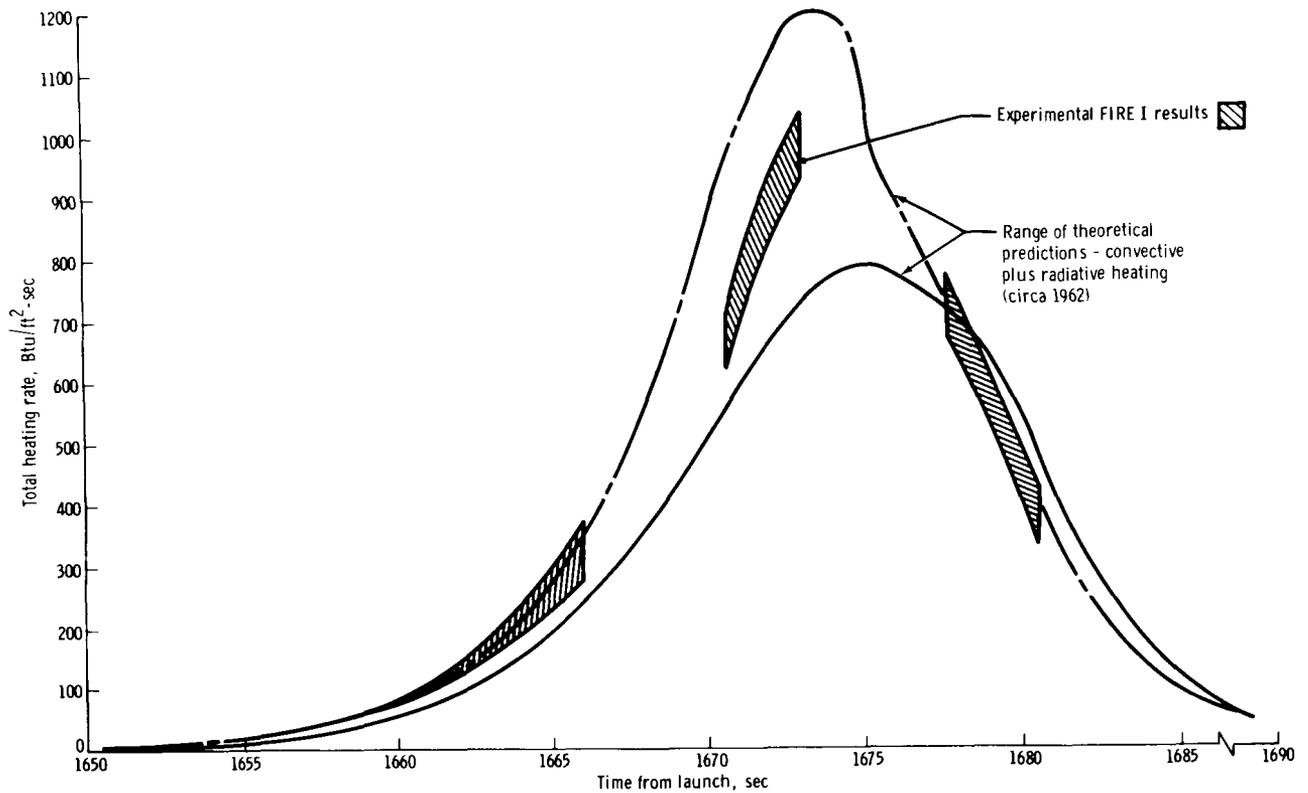


Figure 12. - Measured and predicted heating rates for Project FIRE I during entry.

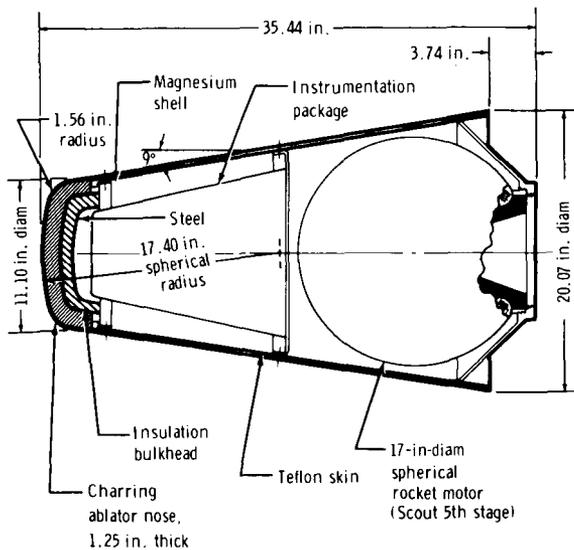


Figure 13. - Spacecraft R-4 ablator material flight test.

much consternation as a result of these findings, and considerable analyses and testing were done to convince the Apollo Program management that the poor performance of the R-4 test spacecraft was a characteristic of the Avco 5026-39 ablator at high aerodynamic pressures, but that the high pressures encountered in the R-4 test were not representative of the Apollo entry environment. From this, it can be concluded that, unless a development flight can be made in an environment representative of the true environment, extraneous issues can arise to cloud the results and cause unnecessary anxiety and work.

### Ablator Backup Program

Late in 1962, concern grew over the increasing weight of the Apollo TPS; it was

hoped that some other ablative filler in a honeycomb matrix might offer a weight advantage. Accordingly, the spacecraft contractor was authorized to conduct a comparative study of the thermal performance and structural properties of five different ablative materials (including the Avcoat 5026-39). The four other materials were the Dow Corning DC-325, which was used on the Gemini spacecraft; a General Electric elastomer designated ESM-1000; the Emerson Electric Thermolag T-500-13; and a silicone material with microballoons and eccospheres that was developed by LRC and was known as purple blend. The results of the study, which lasted 6 months and included extensive testing, showed there was no clear thermal-performance advantage and, consequently, no clear weight advantage to be gained by the use of any one material. In addition, it soon became apparent from the thermostructural tests that the low-temperature requirement of  $-250^{\circ}$  F could not be tolerated by the elastomeric materials and, as a result, the Avcoat 5026-39 was retained as the mainstream material.

By the end of 1963, a renewed concern grew over data that indicated the possibility of an adverse thermal performance of the Avcoat ablator when it was subjected to high aerodynamic pressures. Therefore, another backup program was undertaken for JSC by the Emerson Electric Co., which used a lighter version of the thermolag material (known as T500-111) that had a density of  $35 \text{ lb/ft}^3$ . The material was well characterized and was adaptable to the Apollo manufacturing scheme; however, no worthwhile thermal-performance advantage could be demonstrated. Also, the evaluation of the flight data from the Scout R-4 test in August 1964 finally eliminated the concern over the aerodynamic shear sensitivity of the Avcoat material and the backup program was terminated.

## Boost Protective Cover

Originally, the Block I TPS included approximately 0.12 inch of additional ablator thickness to allow for the charring that would occur during vehicle exit flight. In October 1963, the Apollo Program Manager agreed to a design change that incorporated a boost protective cover over the conical portion of the CM (fig. 5). The boost protective cover was attached to the launch escape tower and was jettisoned with the launch escape tower (ref. 4). The change resulted in the following design improvements.

1. A reduction in ablator thickness and weight
2. Provision of a cover for the CM windows during boost, which eliminated possible contamination of the glass by exhaust products from the tower jettison motors
3. A thermal-control coating that could be applied to the outside surface of the ablator to limit temperature extremes during space flight

The temperature extremes without a coating were of the order of  $\pm 250^{\circ}$  F; and tests had shown that, because of the difference in coefficients of thermal expansion between the ablator and stainless-steel substructure, the ablator cracked if soaked at a temperature less than  $-170^{\circ}$  F.

## Water-Impact Capability

Originally, the Apollo CM was designed to descend on land, and a deployable aft heat shield was attached to the CM primary structure by means of energy-attenuating struts. In the spring of 1964, the primary mode of landing was changed from land to water, and the deploying mechanism for the aft heat shield was deleted because it was not needed for water landings. No modifications were made to the heat-shield substructure to accommodate the water landing because it was believed that this condition would be no more critical than the 20g design entry condition. However, because the analytical prediction of the hydrodynamic pressures and the structural response of the vehicle to these pressures was not understood completely, a qualification test program was planned. The test plan included the dropping of full-scale test articles into a water tank from a pendulum rig to combine horizontal and vertical velocities. The first such test was conducted on October 30, 1964, with a test vehicle designated boilerplate 28. The test conditions simulated a nominal vertical descent on three parachutes ( $V_v = 28$  ft/sec) in combination with a maximum horizontal wind design velocity of 28.5 knots (48 ft/sec). The test resulted in extensive failure of the aft heat shield substructure and of the cabin aft bulkhead, and the test spacecraft sank within 2.5 minutes after contact with the water. The subsequent investigation showed that there was a lack of understanding with respect to the landing criteria and the magnitude of the water-impact pressures. Therefore, the landing criterion was established on a probability basis using a Monte Carlo random selection and a combination of the several variables (wind velocity, wave slope, and so forth). An extensive 1/4-scale-model drop-test program was instituted at the spacecraft contractor facility and at LRC to obtain realistic impact-pressure data. The aft heat shield substructure was redesigned to accommodate the increased pressures, resulting in an increased TPS weight of approximately 230 pounds.

## CONCLUDING REMARKS

The Apollo spacecraft has flown on several earth-orbital and lunar-landing missions, and the thermal protection subsystem has performed well on all missions. The success of the system can be attributed to the conservative design philosophy and rigorous development and verification testing that was conducted. Experience with lunar missions has shown that the spacecraft crewmembers prefer short-range entries and that the command module can be guided precisely to the middle of the entry corridor. This results in a heating environment that is much less severe than that for which the thermal protection subsystem was designed. Entry down range for the manned lunar-landing missions (Apollo 8 to 16) was actually 1500 nautical miles for lunar-return entry, compared with the design requirement of 3500 nautical miles. The resulting maximum integrated heat load was  $26\,500 \text{ Btu/ft}^2$ , compared with  $44\,500 \text{ Btu/ft}^2$  which was used in the design of the thermal protection subsystem.

The following are other major conclusions derived from the thermal protection subsystem design and application experience.

1. The change from the proposed tiled ablator to a monolithic heat shield initiated a lengthy development of a new manufacturing process, and the ablator had to be

modified to permit insertion of the ablator into the honeycomb core by gunning. However, the advantage of having the fail-safe features of a monolithic heat shield embedded in a honeycomb matrix overrode the disadvantages of extended manufacturing schedules and more costly inspection procedures.

2. The down-range distance selected for the Block I design overshoot trajectory was 5000 nautical miles; this distance was reduced to 3500 nautical miles for the Block II design. However, the thermal protection subsystem could not be flight tested beyond 2500 nautical miles, and the astronauts had difficulty ranging greater than 1600 nautical miles down range. As a result, the operational choice has been in favor of ranges less than 1500 nautical miles, which means that the thermal protection subsystem was overdesigned.

3. The various screening and backup programs during the Apollo thermal protection development showed that (a) current ablators of the same general density range (35 to 55 lb/ft<sup>3</sup>) have comparable thermal performance, (b) simple analyses do not form a sound basis for assessing the thermal protection subsystem weight, and (c) flight testing in environments not representative of design conditions, as in the case of the five-stage Scout vehicle with the R-4 payload, can be misleading and cause unwarranted concern if not interpreted correctly.

4. Thermal performance of the ablation material is only one of several criteria required to develop a thermal protection subsystem. Viewing the thermal protection subsystem as a whole, major changes were made in the system to improve inspection access, thermal stress, manufacturing, center-of-gravity control, and performance at singularities; however, major changes in the system were not necessary to obtain better thermal performance from the basic ablator.

5. Because of aerothermodynamic uncertainties associated with the many penetrations, cavities, and protuberances that were required in the heat shield, many of the singularities were recessed into the ablator and other protuberances were located in the leeward regions of separated flow. All of these regions were designed with a fair amount of conservatism.

6. An adequate technology exists to permit the efficient design of ablator thermal protection systems for entry speeds as high as those associated with lunar return. Considerable technological experience has been gained in the design, testing, and analysis of such a thermal protection system. Considering the critical nature of the thermal protection subsystem, the investment of time and money in extensive ground and flight testing is considered to have been worth the effort.

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National Aeronautics and Space Administration  
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914-50-00-00-72

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